Materials Testing on the DC-X and DC-XA

Dane Smith, Carol Carroll, Jochen Marschall, and Joan Pallix

January 1997

National Aeronautics and
Space Administration

Materials Testing on the DC-X and DC-XA

Dane Smith, Carol Carroll, Ames Research Center, Moffett Field, California
Jochen Marschall, Joan Pallix, Thermosciences Institute, Moffett Field, California

Materials Testing on the DC-X and DC-XA

DANE SMITH, CAROL CARROLL, JOCHEN MARSCHALL,* AND JOAN PALLIX*

Ames Research Center

Summary

Flight testing of thermal protection materials has been carried out over a two year period on the base heat shield of the Delta Clipper (DC-X and DC-XA), as well on a body flap. The purpose was to use the vehicle as a test bed for materials and more efficient repair or maintenance processes which would be potentially useful for application on new entry vehicles (i.e., X-33, RLV, planetary probes), as well as on the existing space shuttle orbiters. Panels containing Thermal Protection Systems (TPS) and/or structural materials were constructed either at NASA Ames Research Center or at McDonnell Douglas Aerospace (MDA) and attached between two of the four thrusters in the base heat shield of the DC-X or DC-XA. Three different panels were flown on DC-X flights 6, 7, and 8. A total of 7 panels were flown on DC-XA flights 1, 2, and 3. The panels constructed at Ames contained a variety of ceramic TPS including flexible blankets, tiles with high emissivity coatings, lightweight ceramic ablators and other ceramic composites. The MDS test panels consisted primarily of a variety of metallic composites. This report focuses on the ceramic TPS test results.

Introduction

Flight testing of thermal protection materials has been carried out over a two year period on the base heat shield of the DC-X and DC-XA, as well on a body flap. These activities were funded by Marshall Space Flight Center. The purpose was to use the vehicle as a test bed for materials and more efficient repair or maintenance processes which would be potentially useful for application on new entry vehicles (i.e., X-33, RLV, planetary probes), as well as on the existing space shuttle orbiters to reduce maintenance costs. Panels containing TPS and/or structural materials were constructed either at NASA Ames Research Center or at McDonnell Douglas Aerospace and attached between two of the four thrusters in the base heat shield of the DC-X or DC-XA. Three different panels were flown on DC-X flights 6, 7, and 8. A total of 7 panels were flown on DC-XA flights 1, 2, and 3.

The panels constructed at Ames contained a variety of ceramic TPS including flexible blankets (ref. 1), tiles with high emissivity coatings (refs. 2 and 3), lightweight ceramic ablators (refs. 4–6), and a diboride ceramic matrix composite (refs. 7 and 8). The MDA test panels consisted primarily of a variety of metallic composites. This report focuses on the ceramic TPS test results. Results of the flight tests on metallics can be obtained from Mr. Frank Meyers of MDA.

DC-X Flights, 1995

TPS testing was carried out on DC-X flights 6, 7, and 8 (May 16, June 12, and July 7) in 1995. The most severe thermal environment occurred during vehicle landing. During landing the test panels were exposed to a short duration high temperature heat pulse (peak fluxes up to ~100 W/cm^2) and in most cases were simultaneously impacted by debris from the concrete landing pad. Representative heat flux measurements from calorimeters located in the DC-X and DC-XA base heatshields are shown in figure 1(a). This data was transmitted by Mike Johnson of MDA (Huntington Beach). The heat fluxes measured by calorimeter QC2 are considered the best representation of the heating environment seen by the test panel; see figure 1(b). The initial rise in heat flux corresponds to vehicle lift-off (at about 5 seconds) and the large spikes to vehicle landing. The DC-X flight 6 heating conditions are typical for landings on concrete, which were the case for most of the Delta Clipper flights. The heating conditions for DC-XA flight 1 are for a vehicle landing on a metal grate.

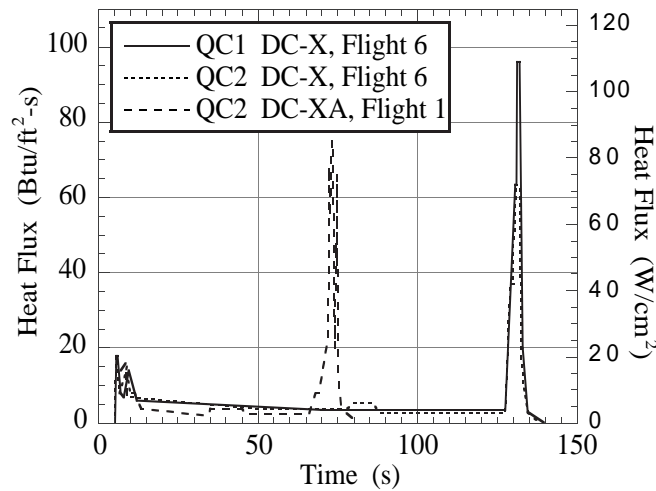*Thermosciences Institute, Moffett Field, CA 94035-1000.

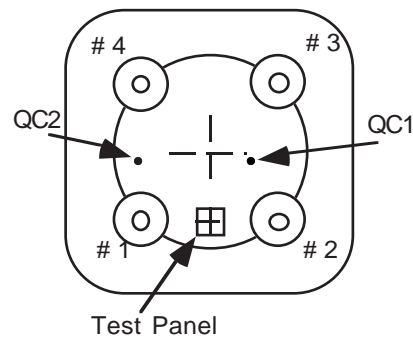Figure 1(a). The approximations to the measured heat fluxes

Figure 1(b). The locations of the two calorimeters relative to the test panel on the base of the DC-X and DC-XA.

Results for Flight 6

The TPS panel tested on flight 6 consisted of four 6 inch × 6 inch × 1 inch tiles of different densities. Two Toughened Unipiece Fibrous Insulation (TUFI) tiles were prepared from 8 lb/ft^3 and 12 lb/ft^3 Alumina Enhanced Thermal Barrier (AETB) ceramic substrates, and are shown in figure 2. TUFI tiles were developed specifically to be resistant to impacts. The white Silicone Impregnated Reusable Ceramic Ablator (SIRCA) tiles in the photo were prepared by silicone impregnation of 9 lb/ft^3 and 18 lb/ft^3 silica tile substrates.
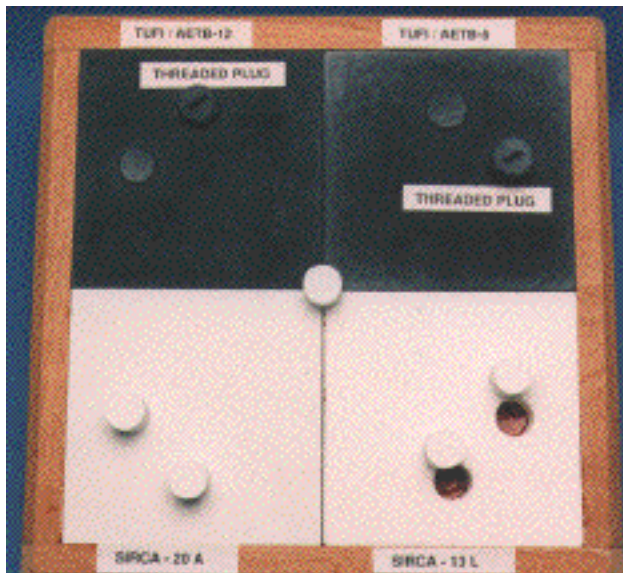

Figure 2(a). Pretest panel for DC-X flight 6. Contains different density TUFI/AETB tiles (black) and SIRCA tiles (white). Note the nine plugs prepared to cover the attachment screws. Two of the plugs were machined out of TUFI so that they could be screwed into place.
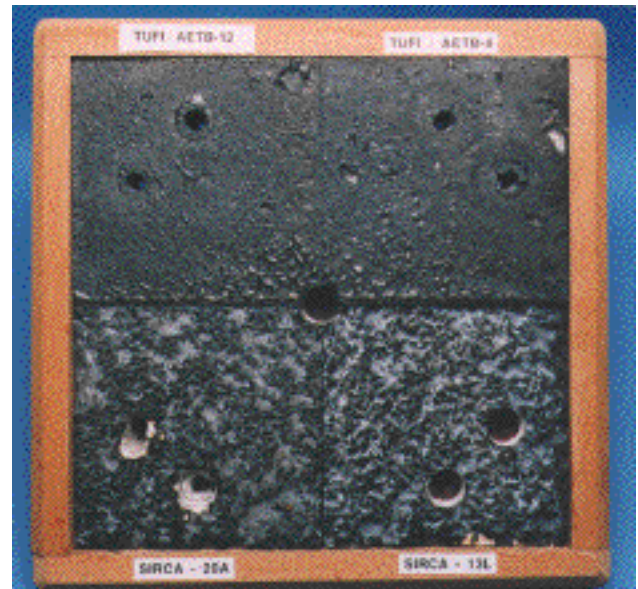

Figure 2(b). Posttest panel shows recession of SIRCA (about 1/4 inch). The plugs have been bored out to remove the panel from the vehicle. Note the textured glaze over the entire panel surface. This is presumably due to deposition of molten concrete. TUFI/AETB-12 sustained the least impact damage.

Figure 2(b) shows the posttest panel. Posttest examination of the panel indicated that the TUFI tiles were essentially unaffected by the high temperature experienced (primarily) during vehicle landing. Both TUFI tiles, however, were pitted due to impact by pieces of solid and molten concrete which spalled from the landing pad. The 12 lb/ft^3 tile sustained much less damage than the 8 lb/ft^3 tile and was patched and reused in the next flight. The SIRCA tiles showed considerable erosion, recession, and some evidence of melting, from the combined thermal and impact environment. Modified versions of SIRCA
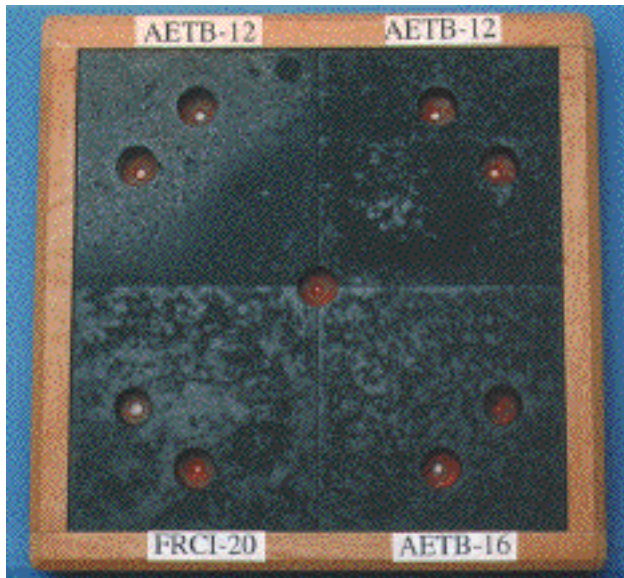
2

Figure 3(a). Pretest panel to be left on the vehicle for DC-X flights 7 and 8. Contains different densities of TUFI/AETB tiles and an FRCI-20 tile. The AETB-12 tile in the upper left corner was already flown on flight 6. FRCI tends to be more durable than AETB but doesn't withstand as high a temperature (ref. 3).

were prepared for flight 7. A layer of glass was deposited in an irregular pattern on all of the TPS surfaces during vehicle landing. This was presumably due to deposition of molten concrete.

Results for Flight 7

In preparation for flight 7, the panel described above was returned to Ames for refurbishing. All of the tiles shown in figure 2 were removed from the panel. Only the TUFI/AETB-12 tile shown in the photo was repaired for additional flight testing.

The other three tiles were replaced with new TUFI tiles including a second TUFI/AETB-12, a TUFI/AETB-16 (16 lb/ft^3 tile) and a TUFI/FRCI-20 tile (20 lb/ft^3 tile; Fibrous Refractory Composite Insulation=FRCI). For this test, the performance of AETB and FRCI was compared. In general, AETB withstands a higher temperature than FRCI but FRCI is inherently stronger (ref. 3). This refurbished panel was left on the vehicle for flights 7 and 8. The pretest panel is shown in figure 3(a).

The panel shown in figure 4 was also flown on flight 7. This panel consisted of a 22 lb/ft^3 SIRCA tile (round white section), a 16 lb/ft^3 organic reusable coating application (ORCA) black semi-circle tile and a protective ceramic coating (PCC) coated SiC Composite Flexible Blanket Insulation (CFBI) black square. The SIRCA and ORCA performed well during this test. The surfaces became much more textured during the flight but there was no significant surface recession and no large
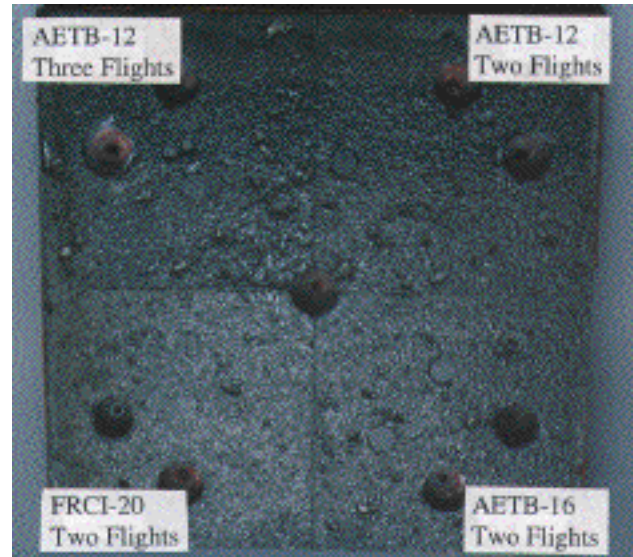

Figure 3(b). Posttest panel after flights 7 and 8. FRCI-20 shows the least impact damage as expected. Note the additional plugs that were installed between flights 7 and 8 in a quick on-vehicle repair.

holes from molten concrete impact (see fig. 4(b)). The CFBI did not perform well and was so badly damaged that only the batting material remained. The mode of failure is not clear from this test. More work was done in 1996 to determine whether failure occurred due to heat flux, aerodynamic loads or impact damage (see discussion of DC-XA tests).
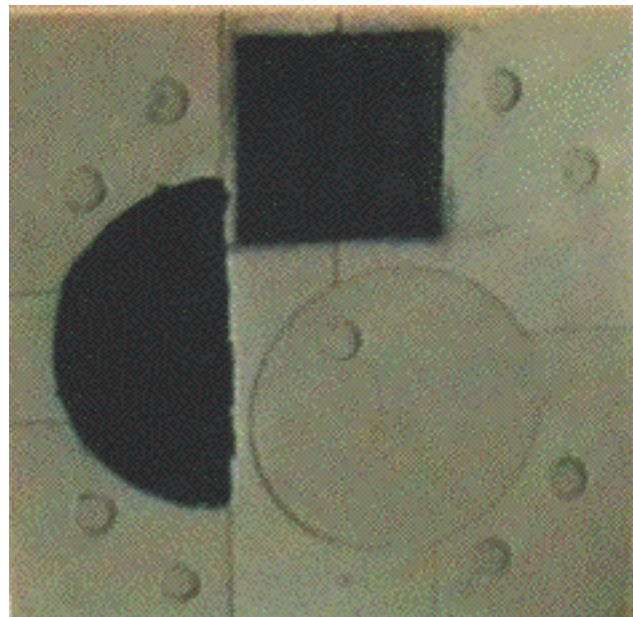

Figure 4(a). Second pretest panel for DC-X flight 7. Contains a variety of forms of SIRCA including a precharred sample (black half circle) which should have a more durable surface. A PCC coated CFBI blanket was also tested on this panel.
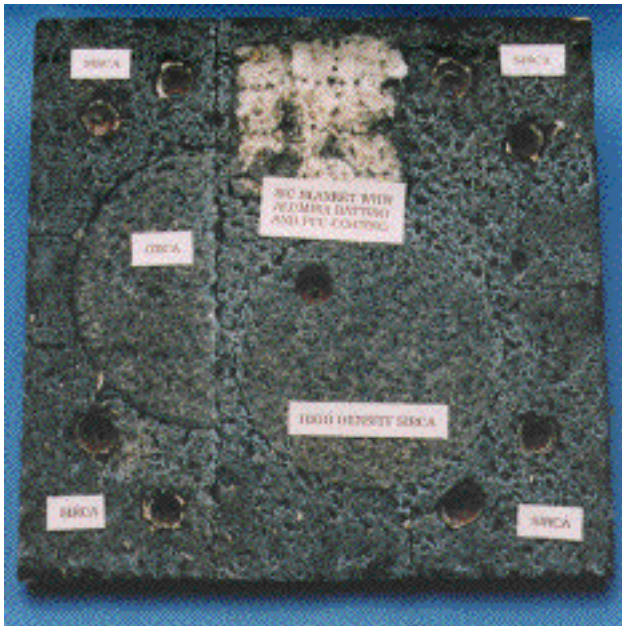
3

Figure 4(b). DC-X flight 7 posttest panel. The surface of the blanket is missing. The high density and precharred SIRCA performed the best. No surface recession was observed in any of the SIRCA samples but the surfaces became textured compared to the preflight specimens.

Results of Flight 8

The panel shown in figure 3(a) was flown again on flight 8 with minor modifications. A technician was sent from Ames to repair several areas on the lower density tiles which had sustained impact damage during flight 7 and to demonstrate an on-vehicle repair process. Four 1/2 inch diameter damaged areas were bored out and plugs were inserted for repair (see fig. 3(b)). The entire repair process took only 15 minutes.

Figure 3(b) shows the flight 8 posttest panel. Minimal additional impact damage was found on any of the TUFI tiles when compared with the previous flight. The actual landing of the DC-X took place much faster than in previous flights so the material exposure to extreme conditions was not as prolonged (i.e., on the order of 4 seconds as opposed to 7 seconds). Figure 3(b) clearly shows that the highest density tiles exhibit the least impact damage as one would expect.

DC-XA Flights, 1996

Materials testing on the base heatshield was carried out on DC-XA flights 1, 2, and 3 (May 18, June 7, and July 31) in 1996. Test panels were fabricated at both Ames and MDA. The MDA test panels consisted of a variety of metallic composites. In addition, a major portion of the base heatshield region was covered with AETB-16 manufactured by McDonnell Douglas Technical Institute

(MDTI). MDA test panel fabrication and testing was organized by Frank Meyers, MDA, and the MDA test results will not be discussed here. The Ames experiments focused mostly on TUFI and SIRCA tiles because these materials performed well on the DC-X. A ZrB_2 specimen (material designation 35v/o SCS-9a-[20v/oRBSiC/ZrB_2]) provided by the Ames Ultra-High Temperature Ceramics (UHTC) group was also tested as a more durable TPS material (refs. 7 and 8).

Results of Flight 1

An Advanced Flexible Reusable Surface Insulation (AFRSI) blanket with two high emissivity coatings applied to the surface was attached to one of the DC-XA body flaps. A black coating, PCC was developed at Ames to improve the thermal properties of AFRSI. A lighter gray area is made up of a proprietary coating developed at Rockwell for the same reason. Blankets placed on the body flap generally experience an aerodynamic load and only minor heating. After the first flight, the blanket appeared to be unchanged but was slightly more rigid than before testing. After the second flight the blanket appeared to be charred but still retained some flexibility. This charring may have been due to a small fire that occurred during landing on this flight. However, no major damage had occurred. The blanket was destroyed during the final DC-XA flight due to the explosion and fire that occurred after a successful flight.

As mentioned previously, a similar blanket did not survive the more extreme conditions of the base heatshield on flight 7 of the DC-X. On the first flight of the DC-XA a more durable blanket was tested on the base heatshield but still did not survive (fig. 5). In this case two

Figure 5(a). DC-XA flight 1 pretest panel. Contains TUFI/ FRCI-20, a ZrB_2 composite, SIRCA tile and silicone impregnated AFRSI blankets with and without precharring.

4

AFRSI blankets (4 inch × 4 inch × 1 inch) were impregnated with silicone which improves the strength of the surface fabric as well as the batting material. The surface of one of the blankets was precharred with a torch prior to testing. The precharring tends to harden the surface which would further improve its durability. It was expected that this material would survive the first DC-XA flight because the vehicle was landing on a metal grate instead of a concrete pad, which would reduce impact damage. Also, the total heat load for this flight was about 60% of that measured for previous landings on concrete. However, after the flight, the front fabric of the blanket was missing and the batting was exposed, just as observed previously for the PCC coated CFBI blanket flown on flight 7 of the DC-X. This was surprising because almost
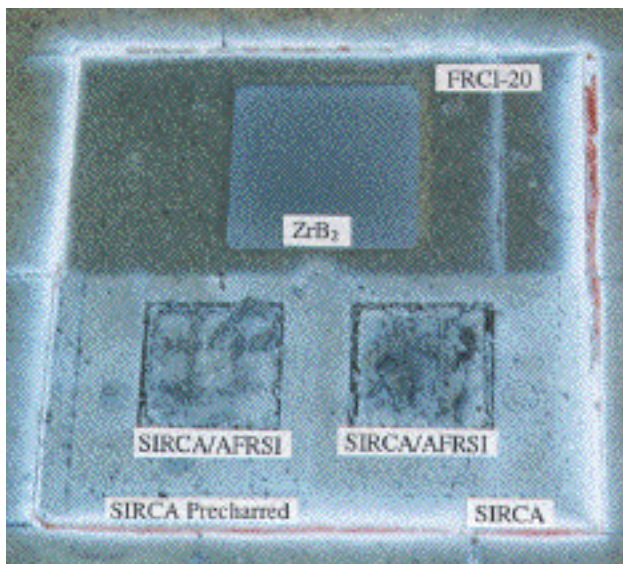

Figure 5(b). DC-XA flight 1 posttest panel. Note that no impact damage has occurred to any of the tiles yet the front fabric of the blankets is missing just as observed a flight 7 of the DC-X.

no impact damage was observed on the materials surrounding the blankets (see fig. 5(b)). Apparently the combined thermal and aerodynamic load experienced by the heat shield is too harsh for AFRSI. If a blanket surface undergoes severe deformation due to aerodynamic loads, the threads used to attach the batting to the fabric will break and the fabric can also tear. It is also likely that the temperature of loose surface fabric will rise sharply, since it has little intrinsic thermal mass and can no longer dissipate heat effectively by conduction to the underlying batting. This may also contribute to blanket failure.

The other materials tested on flight 1 included SIRCA-20, TUFI/FRCI-20 and a ZrB_2 composite. As seen in figure 5(b), no significant damage occurred to any of these materials. However, there were tiny (~$0.5 \times 0.5 \times 0.5$ mm)

pits uniformly distributed over the surface of the SIRCA and no difference was observed for precharred versus uncharred SIRCA. Also, a uniform film had been deposited over the entire panel. This is believed to be due to vaporization of the martite ablative coating that was applied to the landing grate.

Results of Flight 2

Panels tested on DC-XA flight 2 were all fabricated at MDA. Materials consisted of a number of metallic and composite test coupons of various designs. Test results must be obtained directly from MDA.
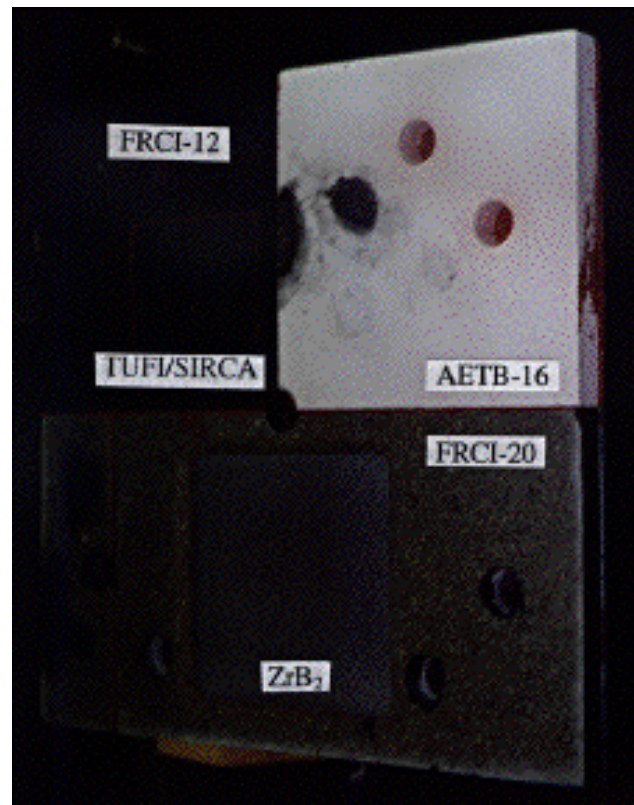

Figure 6(a). DC-XA flight 3 pretest panel. Bottom half of the panel is the one flown on flight 1. On top is a TUFI tile that was impregnated with silicone to add durability. This 4 inch x 4 inch sample is surrounded by a TUFI/FRCI-12 tile. The upper right hand tile is an AETB-16 provided by MDTI. This tile was purposely damaged in the lab to test a quick repair method (see text).

Results of Flight 3

This was the last flight of the DC-XA due to an explosion and fire that destroyed the vehicle. The TPS test panel did survive the fire. The Ames TPS panel tested on flight 3 is shown in figure 6. It consisted of five different TPS materials. A new tile repair process was also tested on this flight and is described in the "processes" section below.
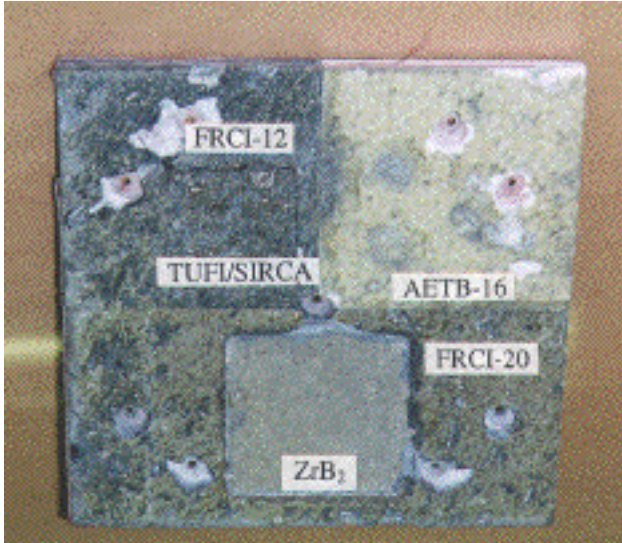
Figure 6(b). DC-XA flight 3 posttest panel. All materials survived and had a silica glaze deposited on the surface during landing on concrete. Note that the bright white areas are damaged due to panel removal from the vehicle and not due to flight damage.

The bottom half of the test panel contained the same TUFI and ZrB2 specimens that were flown on the first flight. This time, the vehicle landed on concrete and the TUFI tile sustained moderate impact damage. The ZrB_2 composite, which is very dense (225 lb/ft^3), was unaffected by the impact or thermal environment. This is the most resilient of the materials tested on any of the DC-X or DC-XA flights.

On the top half of the panel shown in figure 6 there are three different materials including AETB-16 which was provided by MDTI, TUFI coated FRCI-12, and TUFI coated FRCI-20 that was impregnated with silicone. All three materials sustained some impact damage. The white areas around the attachment holes are damage resulting from the tools used to remove the panel after the flight. No photos are available prior to removal of the panel because there was no access to the vehicle during the accident investigation. However, it does appear that impact damage during landing was minimal on the higher density materials (AETB-16 and impregnated TUFI/FRCI-20) and more severe on the TUFI/FRCI-12.

Thermal Analysis

A thermal analysis was carried out to estimate the transient temperature responses of various TPS materials on several flights of the DC-X and the DC-XA. The purpose of this analysis is to estimate the surface and backface temperatures attained during a flight cycle and to compare the thermal responses of various materials under the same heating environment. Computations were made using a one dimensional finite volume heat transfer code developed in house. This code is an earlier version of the One-dimensional Multi-Layer Implicit Thermal Solver (OMLITS) code described in reference 9.

Schematics of the material geometries analyzed are shown in figure 7. The materials involved are the reusable surface insulation (RSI) materials AETB-8, AETB-12,
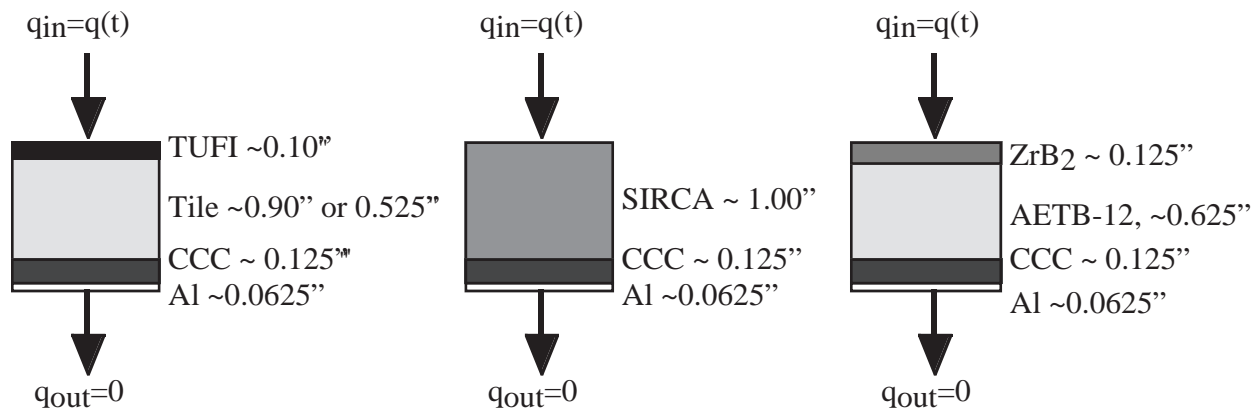

Figure 7. Schematics of the tile, SIRCA, and ZrB$_2$ panels analyzed.

and FRCI-20, the light weight ceramic ablators SIRCA-14 and SIRCA-18, and a diboride ceramic matrix composite. The tile panels consisted of 1 inch or 5/8 inch thick tiles bonded to a 1/8 inch thick carbon-carbon composite (CCC), which was mechanically attached to a 1/16 inch thick aluminum sheet. The AETB and FRCI tiles had a TUFI coating with a thickness was about 0.1 inch (out of the total tile thickness). In practice, an effective thickness was calculated from tile and TUFI densities such that the total added mass per unit area was equal to the manufacturing goal of 0.17 g/cm^2; i.e.,

$$\delta = \frac{0.17 \text{g} \cdot \text{cm}^{-2}}{\rho_{TUFI} - \rho_{RSI}}$$

The SIRCA panels all consisted of 1 inch thick SIRCA bonded to CCC and attached to an aluminum sheet. The ZrB$_2$ sample was 1/8 inch thick and was held on top of a 5/8 inch thick AETB-12 tile by a TUFI/FRCI-20 frame. This tile was bonded to CCC and attached to an aluminum sheet. The thermophysical properties of AETB-8, AETB-12 , FRCI-20 and ZrB$_2$ were obtained from the TPSX data base (ref. 10). The properties of the TUFI coating on the different tiles were communicated by Dave Stewart (Ames). Those for SIRCA-14 and SIRCA-18 were estimated from data obtained by Energy Materials Testing Laboratory for a SIRCA-14 material under reduced pressures (0.05, 0.01, and 0.001 atm). Extrapolation of thermal conductivity data to atmospheric pressures was done using a logarithmic fit to the measured data. Values for SIRCA-18 were further

adjusted (on the order of ±15%) to account for enhanced low temperature and decreased high temperature thermal conductivity, which is expected due to an increase in the conductive contributions and a decrease in radiative contributions to the internal heat transfer. The properties of CCC and aluminum were taken from reference 11.

Contact resistance between layers of materials was neglected and the initial temperature was taken as 293 K. The frontface boundary condition was specified as a time dependent input heat flux and the backface boundary condition (on the aluminum sheet) was made adiabatic. Frontface heatflux was obtained from the QC2 data shown in figure 1. After vehicle landing, the frontface is assumed to cool by radiation exchange with an environment at 293 K.

Surface Temperatures

The computed surface temperatures for the RSI tile test panels on flight 6 of the DC-X are shown in figure 8. The 1 inch and 5/8 inch thick tile specimens show the same surface temperature transients because both are sufficiently thick for very little heat to be drawn from the surface into the CCC and aluminum (over the time scale plotted). The maximum surface temperatures are reached during vehicle touchdown; about 1710 K for AETB-8 at 132 seconds. RSI materials are poor conductors of heat and their surface temperature response is driven by the need to reradiate most of the incoming heat flux. This response is modified in the present case by the thermal mass of the TUFI layer, which damps the rise in surface temperature somewhat. Generally, lower density tile
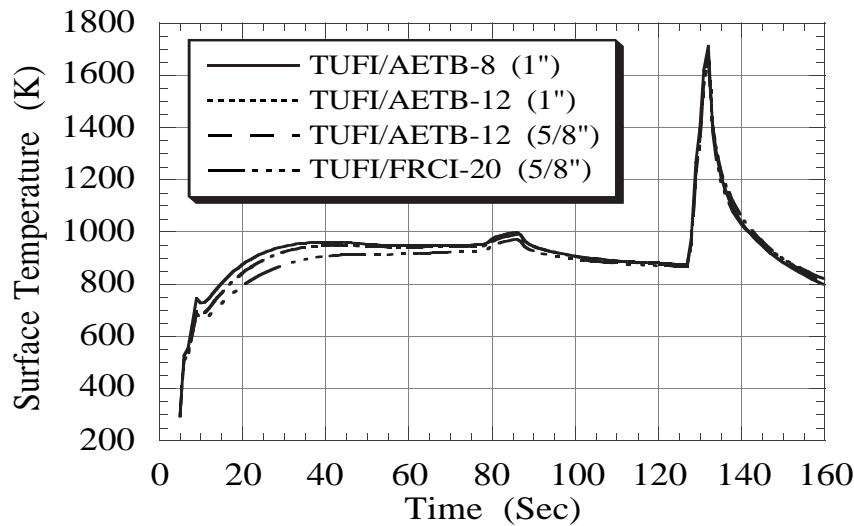

Figure 8. The temporal variation of the surface temperature for four RSI test panels exposed to the heat flux measured by calorimeter QC2 on DC-X flight 6.

materials also have TUFI coatings which are of lower density and which extend further into the material. As a consequence the surface temperature response is fastest on the AETB-8 tile and slowest on the FRCI-20 tile.

The calculated variation of surface temperature on the SIRCA test panels is plotted in figure 9. Again, because heat conduction away from the surface is minimal and surface temperatures are largely controlled by the reradiation of energy which depends on the surface emittance, the temperature variations for SIRCA-14 and SIRCA-18 are almost identical. The surface temperature response to changes in input heat flux is much quicker for SIRCA than for the RSI tile materials because the damping provided by the TUFI layer is not present. The temperature dependent emittance of virgin and fully charred SIRCA has been estimated in reference 5 from room temperature reflectance measurements. The emissivity of the charred material is relatively insensitive to temperature, with values of about 0.95. The emissivity of the virgin material varies greatly, dropping from a room temperature value of about 0.92 to about 0.58 at 1000 K and 0.33 at 2000 K. (These high temperature values for virgin SIRCA assume no char formation at elevated temperatures.) The silicone resin which impregnates SIRCA's silica tile substrate pyrolyzes over a

temperature range of about 500 K to 1000 K. In the present computations the virgin emissivity values were used up to 500 K and the char values were used once the surface temperature had exceeded 1000 K. As the surface temperature increased from 500 K to 1000 K, a linear combination of virgin and charred values was used. Because the surface composition of SIRCA depends upon the particular thermochemical reactions which take place upon heating, it is possible that the emissivity may be lower in some heating environments than in others. The limiting case would be that of negligible char formation, corresponding to the emissivity of a virgin surface. This case is also illustrated in figure 9. The approximate "fail" temperature of silica is 2100 K—around this temperature the material is sufficiently soft that it will flow under aerodynamic forces. The present computations suggest that a charring SIRCA material should survive the DC-X landing environment, reaching a maximum surface temperature of about 1820 K. A virgin surface, however, would not survive, as surface temperatures significantly exceed 2100 K during landing (computations were terminated at 2100 K). The SIRCA-14 and SIRCA-18 test panels which were flown on fight 6 of the DC-X had a charred appearance, yet also showed significant melting, while the neighboring TUFI/AETB-8 and TUFI/AETB-12 panels showed no evidence of melting. This may indicate
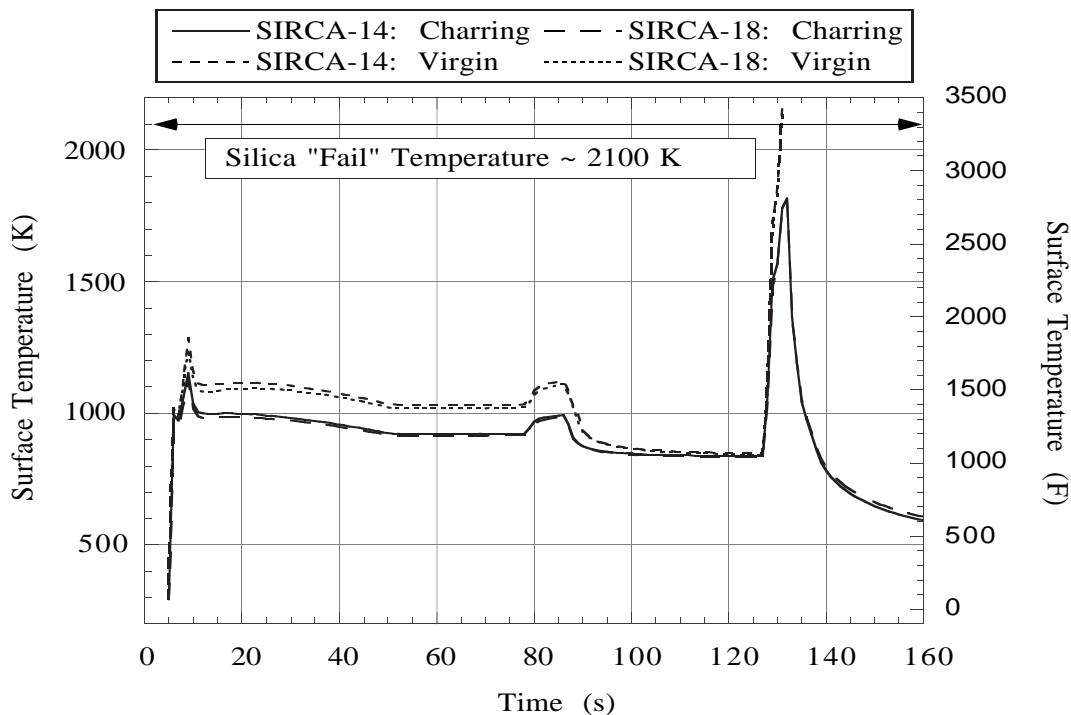

Figure 9. The temporal variation of the surface temperature for SIRCA-14 and SIRCA-18 test panels exposed to the heat flux measured by calorimeter QC2 on DC-X flight 6.
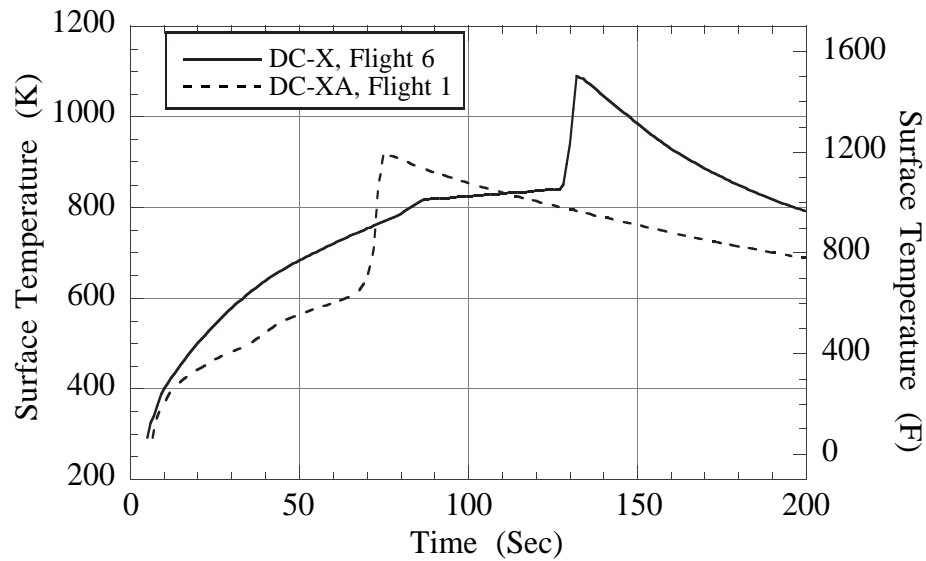
Figure 10. The computed surface temperature of the ZrB_2 specimen for the QC2 calorimeter heat fluxes measured on DC-X flight 6 and DC-XA flight 1.

that the emissivity of the SIRCA samples pyrolyzing in the DC-X heating environment was lower than the ideal char values. Various possible scenarios can be constructed to support this possibility (e.g., the mechanical damage from debris during landing could have exposed virgin material during landing or perhaps the size of the heat pulse during landing was too large and sudden to allow time for char formation before the surface temperature increased past the silica fail temperature), however these are only speculative.

Figure 10 shows surface temperature profiles computed for the ZrB_2 specimen using the heat fluxes measured by calorimeter QC2 on DC-XA flight 1 and DC-X flight 6. (This specimen was only flown on DC-XA flights 1 and 3, however DC-X flight 6 heat fluxes are typical of all landings on concrete and should be representative of conditions experienced on DC-XA flight 3.) The maximum surface temperature reached on the ZrB_2 specimen for the DC-X flight 6 heat flux is about 1100 K. This is hundreds of degrees lower than the temperatures reached by the tile and SIRCA specimens for the same conditions. It is also clear from figure 10 that the temperature response to heat flux variations is much slower than for the tile and the SIRCA samples. Both of

these effects are a direct result of the large thermal mass and relatively high conductivity of the ZrB_2 test specimen. This combination allows the specimen to absorb a large quantity of energy and heat up almost uniformly, with a resulting damped rise in surface temperature.

Flight 1 of the DC-XA was shorter than usual and ended with the vehicle landing on a metal grate to reduce impact damage from spalled concrete. The surface temperature still rises substantially upon landing but the maximum surface temperature reached by the ZrB_2 specimen is lower by almost 200 K. Similar computations for a 5/8 inch thick TUFI/FRCI-20 tile (not shown in fig. 10) also gave lower surface temperatures for DC-XA flight 1 than for DC-X flight 6, though in this case only by about 100 K. This lower surface temperature is due to the lower total heat load because of the shorter flight time and can not be attributed to landing on the grate. However, during flight 1 the engines apparently failed to shut down properly after landing and ran for an additional 2.5 seconds. Thus it is still likely that for identical engine performance and flight times, landing on the grate would contribute to reduced peak surface temperatures when compared to landing on the concrete pad.

9

Table 1. Predicted maximum backface temperatures for different test panels using the heat fluxes measured by QC2 on DC-X flight 6. Thicknesses are for the first material only. The backface temperature is the temperature at the back of the aluminum plate.

TPS material	Thickness (in.)	T_{max} (K)	Time to T_{max} (sec)
TUFI/AETB-8	1.0	354	870
TUFI/AETB-12	1.0	352	1050
TUFI/AETB-12	0.625	381	595
TUFI/FRCI-20	0.625	376	840
SIRCA-14	1.0	337	1220
SIRCA-18	1.0	338	1360
ZrB_2	0.125	402	1265

Backface Temperatures

The maximum calculated temperature rises at the backface of the test panels are listed in table 1. The temperature gradient across the CCC and Aluminum are negligibly small and so the panel backface temperatures are equivalent to the temperatures at the interface of the TPS material and the CCC. In all cases, maximum backface temperatures remained below 410 K. The times given to reach the maximum backface temperature are useful for comparisons among materials, but are somewhat uncertain in the absolute sense because the temperature maxima are very broad in time.

Higher Heat Fluxes

Although the heating environment seen by the test panel is probably best represented by the measurements of calorimeter QC2, it is worth examining how the higher heat fluxes measured by calorimeter QC1 would impact the present analysis. The overall heat load is not significantly different, however the peak heating rate at touchdown is substantially higher for QC1 than QC2.

Numerical computations show the expected result that the backface temperature response is nearly identical for both boundary conditions, while the surface temperatures reach higher values for QC1 heat fluxes than QC2 heat fluxes. The maximum temperature rise for the RSI panels is 1922 K at 132 seconds on the AETB-8 tile. For the SIRCA materials, the surface temperature will exceed 2000 K. Thus, the surface temperature on TUFI/RSI panels remains below the silica fail temperature while the surface temperature of SIRCA will rise very close to the fail temperature. The maximum surface temperature of the ZrB_2 specimen is about 1200 K, which is 100 degrees above the value calculated using the QC2 heat flux boundary condition.

New TPS Processes Tested on the Delta Clipper

Several maintenance and repair processes were tested on Delta Clipper flights which are important to all aerospace programs where ceramic tile technology is applicable. It is important to reduce maintenance costs and manpower in order to ensure rapid vehicle turnaround. Processes tested include direct bonding of tiles and blankets to a composite, quick repair of damaged tiles, and two types of plug inserts.

Panel Attachment

In order to attach test panels to the vehicle, tiles were glued (with RTV 560) to a 12 inch × 12 inch composite which was bolted (9 bolts) to an aluminum plate on the base of the vehicle. A layer of 1/2 inch thick MA25 was between the panel and the aeroshell. The method for direct bonding of the tiles to the composite material was established by preparing the surfaces for bonding and making mechanical measurements of the bond strengths. During the mechanical testing the surface preparation methods were rejected if the test specimen pulled apart at the bonding interface between the tile and the composite. The bonding procedure was considered acceptable when failure occurred in the tile material instead of at the bondline. It was determined that careful cleaning of the bonding surfaces was required so that no dust or other particles remained from machining. It is also important to use a silicone primer on the composite and prepare an RTV transfer coat on the backface of the tiles before bonding. Exact specifications for the bonding procedure can be obtained from Dane Smith at NASA Ames.
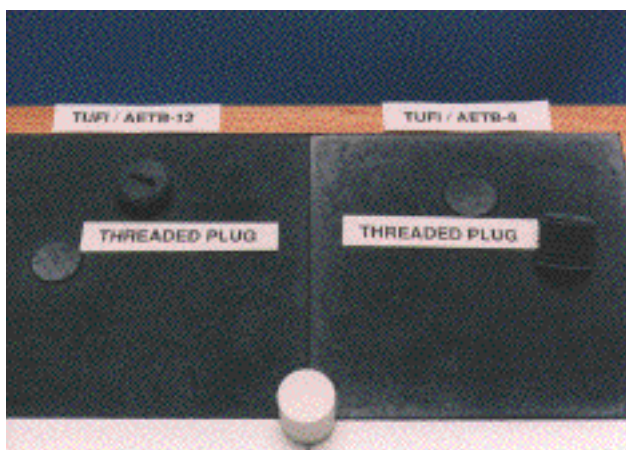
Figure 11. Screws machined from TUFI.

Plugs were fabricated to provide the required protection for the panel attachment screws shown in most of the photos. These were glued in after the composite plate was attached to the vehicle. After testing, the plugs were typically destroyed using a screwdriver to remove the attachment screws. For the first DC-X flight test panel, threaded plugs with a slotted surface (fig. 11) were fabricated for the TUFI panels to demonstrate the machineability of TUFI and to provide reusable plugs which would not require adhesive. During landing, a layer of glass was deposited in an irregular pattern on the TPS surface (fig. 2(b)). This thin glass coating sealed the threaded plugs so that it was necessary to bore them out to remove the panel. The threads inside the holes remained intact so that new threaded plugs could be inserted when necessary.
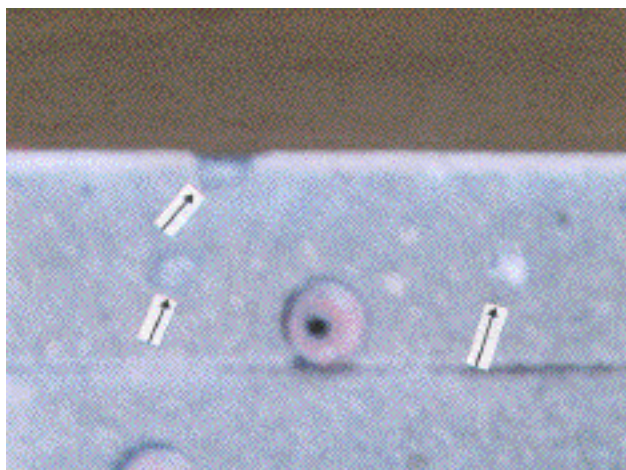

Figure 12(a). Ames fabricated FRCI-20 showing damage in three areas.

Panel Repair

After flight 7 of the DC-X, a low cost technique for patching TUFI tiles was demonstrated with potential application to Shuttle and RLV vehicle operations. Because the ceramic plugs performed very well in the previous testing, the technique was used to repair impact damage on the TUFI tiles (fig. 3). As mentioned previously, a technician did the on-vehicle repair of the damaged TUFI tiles in only 15 minutes and the plugs were intact after DC-X flight 8.

On the last flight of the DC-XA, another new experimental tile repair process was tested which would be even less costly and time consuming than the plug repair process. This "quickfix" consists of a preceramic putty that can be applied to damaged tiles using a spatula and requires no curing. It was tested on both TUFI/FRCI-20 and on an AETB-16 tile that was provided by MDTI. The main purpose of the test is to demonstrate that the putty material is compatible with the ceramic tiles. The putty was formulated to avoid thermal expansion incompatibility. If the thermal expansion of the putty and tiles is very different then the putty may separate from the tile and either leave a gap or fall out completely.

Figure 12 shows the FRCI-20 tile damaged, repaired preflight and repaired postflight. The damage to the FRCI-20 occurred during the first flight of the DC-XA. Damage consisted of two holes approximately 0.5 inches in diameter and 0.25 inches deep. There was also one chipped edge approximately 0.5 inches across and 0.25 inches deep (fig. 12(a)). The damage was repaired using a quickfix containing a high emissivity agent. The repaired tile damage (three black areas) is shown in figure 12(b). After applying the quickfix, a 24-hour drying period was allowed and the patch was as hard as the TUFI. Postflight inspection shows that the patched areas on the surface of
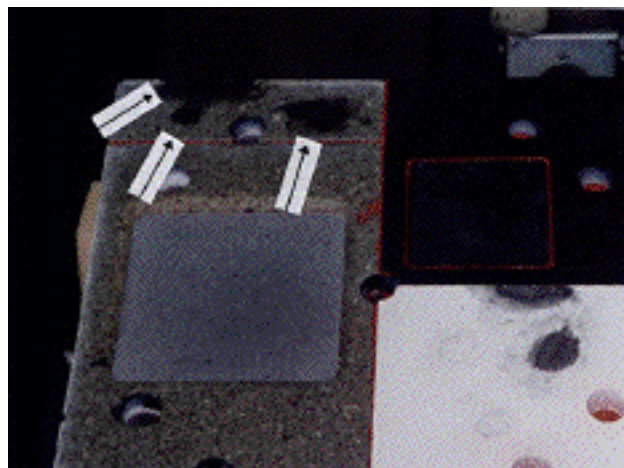

Figure 12(b). FRCI-20 showing repairs using quickfix putty.

the FRCI-20 remained intact while the patch on the edge shows some shrinkage (fig. 12(c)).

Figure 12(c). FRCI-20 posttest. Quickfix adheres well.

The AETB-16 tile was supplied with a high density coating which raised the density to 38 lb/ft^3. The damage areas shown in figure 13(a) were made with a screwdriver just to demonstrate the repair process. There are four damaged areas including two on the surface that are approximately 0.75 inches long and 0.5 inches across and 0.25 inches deep. Another larger area on the surface is about 1 inch in diameter and 0.5 inches deep. The fourth damaged area is along the edge of the tile and is the largest at about 2 inches long and 0.5 inches deep. Figure 13(b) shows that the damaged edge and the large hole were repaired using black quickfix and the two smaller holes were repaired using a white quickfix.
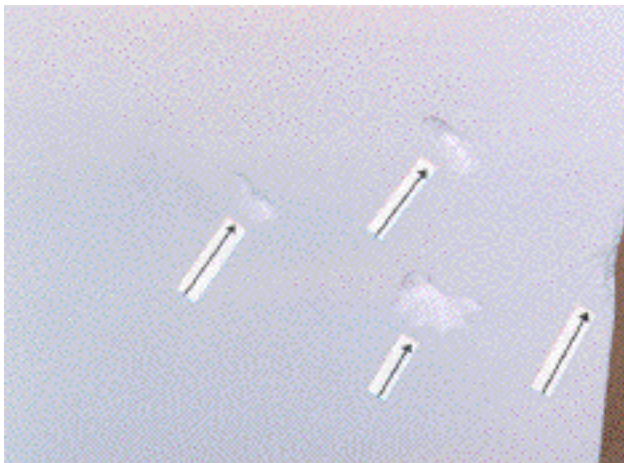

Figure 13(a). AETB-16 provided by MDTI showing damage created with a screwdriver.

On postflight inspection all of the repairs were in good condition with no shrinkage. The white quickfix had turned black and all repairs had provided good protection for the underlying areas. The impact damage in the
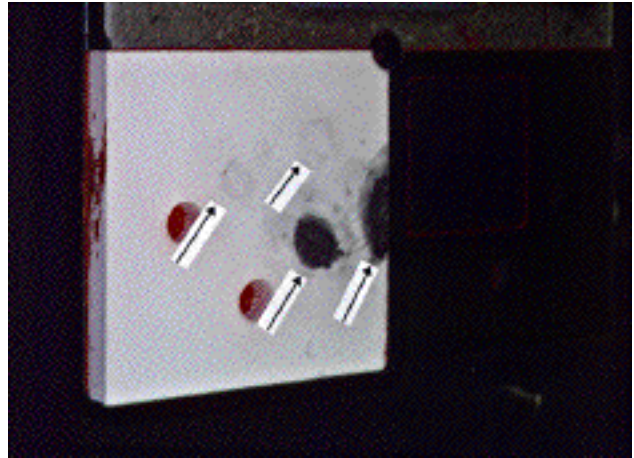

Figure 13(b). MDTI-AETB-16 showing preflight repair using quickfix with and without high emissivity agents.

repaired areas was no worse than on other areas of the tile. Note that the white damage areas seen in figure 13(c) occurred during removal of the tile from the vehicle.
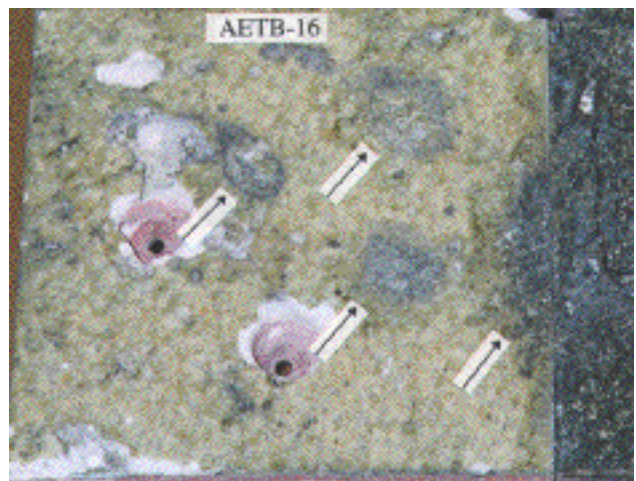

Figure 13(c). MDTI AETB-16 posttest. Shows excellent adherence of quickfix and no severe impact damage to repair areas compared to other areas of the tile.

Conclusions

The diboride ceramic matrix composite was the material that performed the best of all the TPS tested in the Delta Clipper base heatshield experiments. This material was unaffected by thermal or mechanical shock. The material was developed by Advanced Ceramics Research, Tucson, Arizona under NASA Ames funded SBIR contracts for ultra high temperature applications. Although ZrB$_2$ composite is very dense, the material thickness used for these tests was only 0.125 inches. Therefore, it could still be used for some light weight applications or where weight is not a restriction.

A number of ceramic tiles and coatings were tested throughout the Delta Clipper program. Tile densities varied from 8 lb/ft^3 to 20 lb/ft^3 with black and white coatings. All of the tiles appeared to be unaffected by the thermal environment but sustained damage due to debris impact during vehicle landing. The least damage was incurred on the highest density materials (i.e., TUFI/FRCI-20 and silicone impregnated TUFI/FRCI-20).

Low density (8 and 12 lb/ft^3) SIRCA test specimens did not do well in the severe thermal and high impact environment. The higher density SIRCA was much more durable but the postflight surfaces were much more textured than preflight surfaces. It is expected that proper precharring of the surface would improve its strength and reduce this effect. Precharred samples were prepared but never tested due to the premature end of the Delta Clipper program.

Base heatshield testing of blankets were complete failures in every case. The only blanket that did well was the one placed on a body flap. The only adverse effects observed on the body flap test specimen occurred as a result of fire on the second flight (although the blanket was destroyed in the fire on the final flight).

All of the maintenance and repair processes tested were extremely successful. The technique developed to bond test materials directly to a composite never failed throughout all of the flight testing. Several successful tile repair processes were demonstrated which would significantly reduce costs and manpower required to maintain rapid turnaround vehicles.

References

1. Kourtides, D.A.; Chiu, S.A.; Iverson , D.J. and Lowe, D.M.: Thermal Response of Rigid and Flexible Insulations and Reflective Coating in an Aeroconvective Heating Environment. NASA TM-103925 (1992); U.S. Patent No. 5,038,693 (1991).

2. Leiser, D.B.; Churchward, R.; Katvala, V.; Stewart, D. and Balter, A.: Advanced Porous Coating for Low-Density Ceramic Insulation Materials. J. Am. Ceram. Soc., 72(6), 1003-1010, 1989; U.S. Patent No. 5,079,082, 1992.

3. Leiser, D.B.; Smith, M.; and Stewart, D.A.: Options for Improving Rigidized Ceramic Heatshields. Ceram. Eng. Sci. Proc., vol. 6, no. 757, 1985.

4. Tran, H.; Johnson, C.; Rasky, D.; Hui , F.; and Hsu, M.: Silicone Impregnated Reusable Ceramic Ablators for Mars Follow-on Missions. AIAA 96-1819, 31st AIAA Thermophysics Conf., June 17-20, 1996, New Orleans, Louisiana.

5. Tran, H.; Rasky, D.J. and Esfahani, L.: Thermal Response and Ablation Characteristics of Light-Weight Ceramic Ablators. AIAA 93-2790, AIAA 28th Thermophysics Conf., July 6-9, 1993, Orlando, Florida.

6. Tran, H.K.: Development of Lightweight Ceramic Ablators and Arc-Jet Test Results. NASA TM-108798, January 1994.

7. Stuffle, K.; and Bull, J.D.: Continuous Fiber Reinforced Composites for Heat Shield Applications Proceedings of the Conference on Processing, Fabrication and Application of Advanced Composites. Long Beach, California, pp. 63-72, August 9-11, 1993.

8. Work performed in NASA Ames funded SBIR Phase II, Contract Number NAS2-13896.

9. Milos, F.S.; Chen, Y.-K.; and Henline, W.D.: Methodology for Full-Body TPS Sizing of Access-to-Space Vehicles. AIAA 96-0614, 34th AIAA Aerospace Sciences Meeting, January 1996, Reno, Nevada.

10. TPSX — Thermal Protection Systems Expert and Materials Properties Database, Version 1.1, October 1996.

11. Williams, S.D.; and Curry, D.M.: Thermal Protection Materials; Thermophysical Property Data. NASA Reference Publication 1289, 1992.

REPORT DOCUMENTATION PAGE

1. AGENCY USE ONLY *(Leave blank)*	2. REPORT DATE January 1997	3. REPORT TYPE AND DATES COVERED Technical Memorandum

4. TITLE AND SUBTITLE

Materials Testing on the DC-X and DC-XA

5. FUNDING NUMBERS

242-30-01

6. AUTHOR(S)

Dane Smith, Carol Carroll, Jochen Marschall,* and Joan Pallix*

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

Ames Research Center
Moffett Field, CA 94035-1000

8. PERFORMING ORGANIZATION REPORT NUMBER

A-975493

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

NASA TM-110430

11. SUPPLEMENTARY NOTES

Point of Contact: Joan Pallix, Ames Research Center, MS 200-1, Moffett Field, CA 94035-1000
(415) 604-0332
*Thermosciences Institute, Moffett Field, CA 94035-1000

12a. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified-Unlimited
Subject Category –23

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

Flight testing of thermal protection materials has been carried out over a two year period on the base heat shield of the Delta Clipper (DC-X and DC-XA), as well on a body flap. The purpose was to use the vehicle as a test bed for materials and more efficient repair or maintenance processes which would be potentially useful for application on new entry vehicles (i.e., X-33, RLV, planetary probes), as well as on the existing space shuttle orbiters. Panels containing Thermal Protection Systems (TPS) and/or structural materials were constructed either at NASA Ames Research Center or at McDonnell Douglas Aerospace (MDA) and attached between two of the four thrusters in the base heat shield of the DC-X or DC-XA. Three different panels were flown on DC-X flights 6, 7, and 8. A total of 7 panels were flown on DC-XA flights 1, 2, and 3. The panels constructed at Ames contained a variety of ceramic TPS including flexible blankets, tiles with high emissivity coatings, lightweight ceramic ablators and other ceramic composites. The MDS test panels consisted primarily of a variety of metallic composites. This report focuses on the ceramic TPS test results.

14. SUBJECT TERMS Thermal production, Ceramic TPS, DCX	15. NUMBER OF PAGES 16
	16. PRICE CODE A03

17. SECURITY CLASSIFICATION OF REPORT Unclassified	18. SECURITY CLASSIFICATION OF THIS PAGE Unclassified	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT